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TECHNICAL NOTE

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COMPARISON OF TWO MANEUVERS FOR LONGITUDINAL

RANGE CONTROL DURING ATMOSPHERE ENTRY

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SUMMARY

An analysis has been made of atmosphere entry trajectories to determine the range for two types of maneuvers. For one type of maneuver range was controlled principally by varying the point in the trajectory where the lift-drag ratio was reduced to maximum negative lift-drag ratio. For the other type of maneuver range was controlled principally by varying the value to which the lift-drag ratio was reduced. The influence on range of maximum deceleration limits and an error in lift-drag ratio was included; the convective and radiative heating to the stagnation point of the spacecraft were also studied. The analysis was made for a spacecraft with a maximum lift-drag ratio of 0.5 entering the earth's atmosphere at parabolic velocity. The results for both types of maneuvers indicate that the spacecraft aerodynamics can provide ranges up to global range. For an error in lift-drag ratio of ±0.001 over part of the trajectories, the range is obtained with relatively small errors for one of the types of maneuvers, depending upon deceleration limit and corridor depth. Essentially, no penalty in heating is incurred for ranges from about 5,000 miles to global range.

INTRODUCTION

In the return of a manned spacecraft to earth, the ability to control the trajectory and land at a predesignated area or areas is of major importance. A factor of significance in achieving this goal is the control of range from the entry to touchdown. There have been a number of investigations (see, e.g., refs. 1 through 6) to assess this range control problem for manned spacecraft. Generally, in these investigations, maneuvers to control the range were studied for a single deceleration limit; the studies were made without considering the accuracy requirements and capabilities of the spacecraft in sensing and controlling lift-drag ratio during entry.

In the present investigation two types of maneuvers to control the range are studied with consideration given to errors in lift-drag ratio. For one type of maneuver, various deceleration limits are considered. The maneuvers consist of discrete changes in lift-drag ratio obtained by changes in angle of attack of a spacecraft with an assumed drag polar. The errors in lift-drag ratio could be due to guidance system errors in sensing deceleration and flight-path angle, or to errors in controlling the spacecraft. With the control of range, the question logically arises as to the magnitude of the attendant heating. Accordingly, convective and radiative heating results are also described. To obtain these results a trajectory analysis has been made which uses an IBM 7090 computing machine to solve the two-dimensional equations of motion of references 7 and 8. The results obtained are for parabolic entry velocity, a non-rotating spherical earth, and an exponential atmosphere. The spacecraft has a maximum lift-drag ratio of 0.5 and follows trajectories that either stay within the sensible atmosphere or skip out of the atmosphere but stay below an altitude of 400 statute miles.

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NOTATION

A reference area for drag and lift, ft^2

C_D drag coefficient, $\frac{2D}{\rho V^2 A}$

C_{D_o} drag coefficient at zero lift

C_L lift coefficient, $\frac{2L}{\rho V^2 A}$

D drag force, lb

g local gravitational acceleration, ft sec^{-2}

G deceleration in g units

L lift force, lb

m mass of vehicle, slugs

q total heat absorbed at the stagnation point, Btu ft^{-2}

$\dot{q}$ heating rate at the stagnation point, $\frac{dq}{dt}$, Btu $ft^{-2} sec^{-1}$

r distance from the center of planet, ft

r_o radius of planet, 2.0926×10^7 ft for earth

R radius of curvature of spacecraft surface, ft

s range, ft

t time, sec

u tangential velocity component normal to a radius vector, ft sec^{-1}

v radial velocity component, ft sec^{-1}

V resultant velocity, ft sec^{-1}

y altitude, ft

α angle of attack of spacecraft, deg

A597

β atmosphere density decay parameter, 1/23,500 ft^{-1} for earth

γ flight-path angle relative to the local horizontal, negative for descent, deg

μ gravitational constant, 1.4078×10^{16} $ft^3 sec^{-2}$ for earth

ρ atmosphere density, slugs ft^{-3}

ρ_o atmosphere density at planet surface, 0.00238 slug ft^{-3} for earth

$\bar{\rho}_o$ mean value for exponential approximation to atmosphere density-altitude relation, 0.0027 slug ft^{-3} for earth

Subscripts

c convective

i initial

max maximum

p vacuum perigee

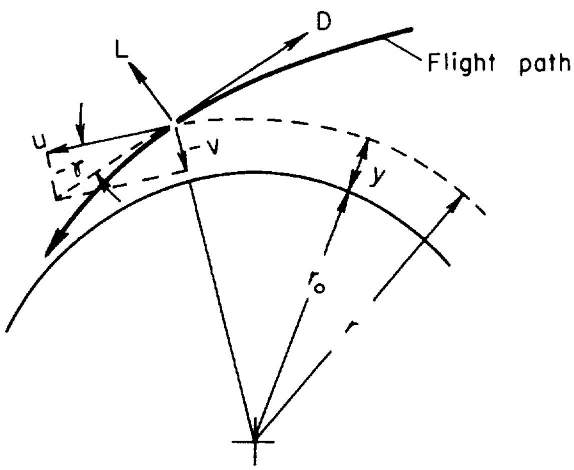
r radiative

ANALYSIS

Most of the analysis is identical to that in reference 7 and is repeated here for the convenience of the reader.

Trajectory Equations

A trajectory analysis has been made utilizing the solution of two-dimensional equations of motion for entries into an exponential atmosphere of a nonrotating spherical earth. The polar coordinate system with velocity components, aerodynamic forces, and flight-path angle is defined in the sketch. The differential equations for the velocity in the radial and tangential directions are, respectively, (see ref. 8)



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$$\frac{dv}{dt} = -g + \frac{u^2}{r} + \frac{L}{m}\cos\gamma - \frac{D}{m}\sin\gamma \qquad (1)$$

$$\frac{du}{dt} = -\frac{uv}{r} - \frac{D}{m}\cos\gamma - \frac{L}{m}\sin\gamma \qquad (2)$$

where

$$\tan\gamma = \frac{v}{u} \qquad (3)$$

$$r = r_o + y \qquad (4)$$

and g is the local gravitational acceleration given by

$$g = \frac{\mu}{r^2} \qquad (5)$$

The constant μ in equation (5) is the gravitational constant defined by Newton's inverse square law of gravitational attraction. The differential equations employed for the altitude and range are, respectively,

$$\frac{dy}{dt} = v \qquad (6)$$

$$\frac{ds}{dt} = u \qquad (7)$$

The differential equations used for the total laminar convective heat (after ref. 8) and the radiative heat (ref. 9) absorbed per unit area at the stagnation point are, respectively,

$$\frac{dq_c}{dt} = \dot{q}_c = \frac{16,600}{\sqrt{R}}\sqrt{\frac{\rho}{\rho_o}}\left(\frac{V}{\sqrt{gr}}\right)^3 \qquad (8)$$

$$\frac{dq_r}{dt} = \dot{q}_r = R10^{f(y,V)} \qquad (9)$$

where R is the radius of curvature of the spacecraft surface at the stagnation point, ρ is the local atmosphere density given by

$$\rho = \bar{\rho}_o e^{-\beta y} \tag{10}$$

and V is the resultant velocity given by

$$V = \sqrt{u^2 + v^2} \tag{11}$$

The radiative heating rate per unit area at the stagnation point is obtained from an interpolation of a table for the logarithm of equation (9), that is,

$$\log_{10}\left(\frac{\dot{q}_r}{R}\right) = f(y,V) \tag{12}$$

Values of $(\dot{q}_r/R)$ as a function of altitude and velocity for air in equilibrium were obtained from reference 9. The six equations (1), (2), (6), (7), (8), and (9) were programed for simultaneous solution on an IBM 7090 computing machine.

Spacecraft Characteristics

Modulation is accomplished in the present investigation by varying the spacecraft drag coefficient and lift-drag ratio during entry. The drag coefficient and lift-drag ratio are calculated on the assumption (as in ref. 10) that the spacecraft has a variation of lift and drag similar to that for a flat plate in Newtonian flow given by

$$C_D = C_{D_o} + \left(C_{D_{max}} - C_{D_o}\right) \sin^3\alpha \tag{13}$$

$$C_L = \left(C_{D_{max}} - C_{D_o}\right) \sin^2\alpha \cos\alpha \tag{14}$$

$$\frac{L}{D} = \frac{\sin^2\alpha \cos\alpha}{b + \sin^3\alpha} \tag{15}$$

where

$$b = \frac{C_{D_o}}{C_{D_{max}} - C_{D_o}}$$

The particular drag polar with a maximum lift-drag ratio of 0.5 used in the present investigation is given in figure 1. The results, however, can be shown to apply to a family of spacecraft having $(L/D)_{max} = 0.5$ with various values of $C_{D_{max}}$ and C_{D_o} provided $C_{D_{max}}/C_{D_o}$ remains constant and the initial m/A is adjusted to give initial values of m/C_DA equal to those used in the present investigation. A value of $m/A = 3$ was used in the present calculations.

The results of the present analysis can be applied to spacecraft of arbitrary weight and size by employing the results of reference 8. According to these results, many of the trajectory parameters are essentially independent of m/C_DA. In this category are deceleration, velocity, flight-path angle, and range. Trajectory parameters that depend on m/C_DA are altitude and convective and radiative heating where the relationships for altitude and convective heating can be shown to be

$$y_2 - y_1 = \frac{1}{\beta} \log_e\left[\frac{(m/C_DA)_1}{(m/C_DA)_2}\right] \tag{16}$$

and

$$\frac{q_{c_2}}{q_{c_1}} \sim \frac{\dot{q}_{c_2}}{\dot{q}_{c_1}} \sim \sqrt{\frac{(m/C_DAR)_2}{(m/C_DAR)_1}} \tag{17}$$

The subscript 1 refers to values of the present report and the subscript 2 corresponds to other values of m/C_DA or m/C_DAR. Equations (9) and (16) and reference 9 can be used to calculate radiative heating if the velocity-time and velocity-altitude relationships are given.

Corridor Depth

In general, corridor depth is defined as a difference in perigee altitudes of vacuum trajectories corresponding to a difference in two flight-path angles at a given initial altitude and velocity. The flight-path angle associated with the higher of the two perigee altitudes is referred to as the angle for an overshoot boundary. In the present paper this angle is defined as the most shallow angle for which the spacecraft will not skip beyond an altitude of 400 statute miles after entering at its highest negative lift-drag ratio. For the spacecraft studied in this paper, $m/A = 3$, $(L/D)_{max} = 0.5$, this angle is determined to be -5.03° for an initial altitude of 400,000 feet and parabolic velocity. The angle associated with the lower of the two perigee altitudes is referred to as the angle for an undershoot boundary. This angle can be defined as the steepest angle for which the spacecraft enters the atmosphere without exceeding specified constraints of deceleration, heat loads, Reynolds number, etc. (see, e.g., refs. 10 and 11). In the present paper the angle for an undershoot boundary is determined by only a deceleration limit; deceleration limits from about 2 to 10 are considered. After peak deceleration has been reached, the spacecraft is maneuvered both to stay within and to skip beyond the sensible atmosphere with the stipulation that subsequent decelerations do not exceed the peak value first experienced. The relationship between perigee altitude and flight-path angle for a spacecraft at parabolic velocity can be shown from vacuum trajectory relationships to be

$$y_p = y_i\cos^2\gamma_i - r_o(1 - \cos^2\gamma_i) \tag{18}$$

Perigee altitude and corridor depth are presented in figure 2 as functions of flight-path angle for an initial altitude of 400,000 feet. A corridor depth of zero corresponds to entry along the overshoot boundary, $\gamma_i = -5.03^\circ$.

RESULTS AND DISCUSSION

Atmosphere entry trajectories are analyzed to determine the range for two types of maneuvers which involve discrete changes in lift-drag ratio. For one type of maneuver, range is controlled principally by varying the point in the trajectory where the lift-drag ratio is reduced to maximum negative lift-drag ratio. For the other type of maneuver, range is controlled principally by varying the value to which the lift-drag ratio is reduced. The influence on range of maximum deceleration limits and an error in lift-drag ratio is determined; the convective and radiative heating also are studied. The error in lift-drag ratio employed, 0.001, is calculated by use of equations (1) and (2) with an assumption of currently realistic guidance system errors of 10^{-4}g in measuring deceleration and 0.05° in measuring flight-path angle.

Comparison of the Two Types of Maneuvers

The altitude-range relationships for the two types of maneuvers are shown in figure 3 for the same initial flight-path angle ($\gamma_i = -7.52^\circ$). The entries start with the spacecraft at maximum lift-drag ratio ($L/D = 0.5$). For the entry shown by the solid curve, the initial L/D is held constant until after peak deceleration and only long enough so that an instantaneous change to $L/D = -0.5$ prevents the spacecraft from skipping higher than 300,000 feet. At the top of the skip the L/D is changed to +0.5 to obtain maximum range from this point to touchdown. For the entry shown by the dashed curve, the initial L/D is held constant until the bottom of the pull-up, $\gamma \approx 0^\circ$. Then the L/D is reduced only as much as is required to prevent the vehicle from skipping beyond 300,000 feet, to a value of 0.07 for the entry shown. At the top of the skip the L/D is again changed to +0.5. For both entries, the first changes in L/D are made in a region of relatively high dynamic pressure where any change in L/D greatly affects the subsequent trajectory. To assess the sensitivity of range to errors in L/D in this region, the trajectories are also calculated with $L/D = -0.5 + 0.001$ and $L/D = 0.07 \pm 0.001$ from the points indicated to the tops of the subsequent skips. It can be seen that initiating the reduction in L/D

at the bottom of the pull-up, rather than using maximum L/D for a longer time, results in a decrease in range of about 1700 miles. It should be emphasized that for the entries shown, the vehicle is required to stay within the sensible atmosphere; that is, the desired trajectory calls for a skip altitude of 300,000 feet. When an error in L/D of ±0.001 is introduced in the entry at the bottom of the pull-up, the skip altitude of the actual trajectory varies only a few thousand feet above or below that for the desired trajectory and the resulting variation in range is about 200 miles. However, when an error in L/D of 0.001 is introduced in the entry using maximum L/D after $\gamma = 0$, the spacecraft actually skips to an altitude of somewhat less than 500,000 feet and the resulting uncertainty in range is about 6,600 miles. The larger uncertainties in range, of course, require more corrective maneuvers to reach a desired landing area.

The uncertainty in range due to an error in L/D, when the vehicle is allowed to skip to desired altitudes between 300,000 feet and 400 miles to obtain long range, is shown in figure 4. Uncertainty in range is shown as a function of range for the two types of entries described previously. For the entries with the maximum L/D held constant until γ is greater than 0° (solid curve), the range is controlled primarily by varying the time of application of maximum positive L/D. For the entries with the maximum L/D held constant only until the bottom of the pull-up (dashed curve), the range is controlled primarily by varying the L/D at the bottom of the pull-up. For these latter entries, the uncertainty in range is relatively small, even for ranges over 20,000 statute miles; and the uncertainty in range is relatively insensitive to range. For the entries with the maximum L/D held constant until γ is greater than 0°, the uncertainty in range decreases rapidly with increasing range but it is always greater than that for the entries with the maximum L/D held constant only until the bottom of the pull-up.

Influence of Maximum Deceleration on Range

The influence of maximum deceleration on range and uncertainty in range is shown in figure 5. The results are presented for entries during which $(L/D)_i$ is held constant only until the bottom of the pull-up.[1] These results are for desired skip altitudes of 300,000 feet and 400 miles for corridor depths of 10, 20, and 30 miles. The cross-hatched boundaries indicate the entries which require the maximum initial L/D capability of the vehicle ($(L/D)_i = 0.5$) to obtain the lowest possible deceleration. Higher decelerations are a result of decreased initial L/D. For each corridor depth, the uncertainty in range resulting from an error in L/D is indicated by the dotted areas. For example, as shown in figure 5(a), for entry along the 5.6g-limited undershoot boundary of a 20-mile corridor the range varies from about 16,000 miles to 8,000 miles for an

[1]At the bottom of the pull-up the L/D is reduced to obtain the desired skip altitude. The L/D is then changed to $(L/D)_{max}$ to obtain maximum range from this point to touchdown.

error in L/D of ±0.001. It should be recalled that the introduction of an error in L/D results in an actual skip altitude which deviates from the desired value. The results presented in figure 5(a) for a desired skip altitude of 300,000 feet show that for maximum decelerations from about 8g to 10g, ranges on the order of 6,000 miles are obtained with relatively little uncertainty for an error in L/D of ±0.001. The midcourse guidance requirements are not considered stringent since it is necessary for the spacecraft to hit corridor depths of not less than about 20 miles (see, e.g., ref. 12). For maximum decelerations of about 5g, however, the uncertainty in range is extremely large unless the midcourse guidance system can guide the spacecraft to very small corridor depths. Even for a corridor depth of 10 miles, the uncertainty in range is about 2500 miles.

The results presented in figure 5(b) for a desired skip altitude of 400 miles are similar to those for skips limited to 300,000 feet. However, for maximum decelerations from about 8g to 10g ranges over 20,000 miles can be obtained with relatively little uncertainty for corridor depths greater than 20 miles. For a maximum deceleration of 5g, global range can be obtained with little uncertainty for corridor depths of 10 miles or less.

Stagnation-Point Heating

The stagnation-point heating of the spacecraft associated with variations in range and maximum deceleration is shown in figure 6. The spacecraft is assumed to have a spherical face with a radius of 10 feet and an m/A of 3. The total convective heating per unit area (on the left) and the total radiative heating per unit area (on the right) are shown as functions of the range. The maximum heating rates per unit area are also indicated. The results are shown for undershoot boundaries limited by maximum decelerations of 5g and 10g, and for maneuvers in which the initial L/D is held constant until $\gamma \approx 0^{\circ}$. It can be seen that the total convective and radiative heating are relatively insensitive to range. This is a result of the fact that most of the heating occurs at relatively low altitudes, whereas most of the range is obtained at high altitudes. Thus, essentially no penalty in heating is paid for obtaining the longer ranges, even for those as long as global range. The total convective heating is higher for the entries limited by maximum decelerations of 5g than 10g; the reverse is true for the total radiative heating and the heating rates. The results are essentially the same for the entries in which the initial L/D is held constant as long as possible.

CONCLUDING REMARKS

Atmosphere entry trajectories have been computed to determine the range for two types of maneuvers. The influence on range of maximum deceleration limits and an error in lift-drag ratio have been determined; the convective and radiative heating also were studied. The study was made for a spacecraft with a maximum lift-drag ratio of 0.5 entering the earth's atmosphere at parabolic velocity. The results indicate that the spacecraft aerodynamics can be controlled for both types of maneuvers to obtain ranges up to global range. For an error in lift-drag ratio of ±0.001 over part of the trajectories, the range is obtained with relatively small errors for one of the types of maneuvers, depending upon deceleration limit and corridor depth. Essentially, no penalty in heating is incurred for ranges from about 5,000 miles to global range.

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National Aeronautics and Space Administration
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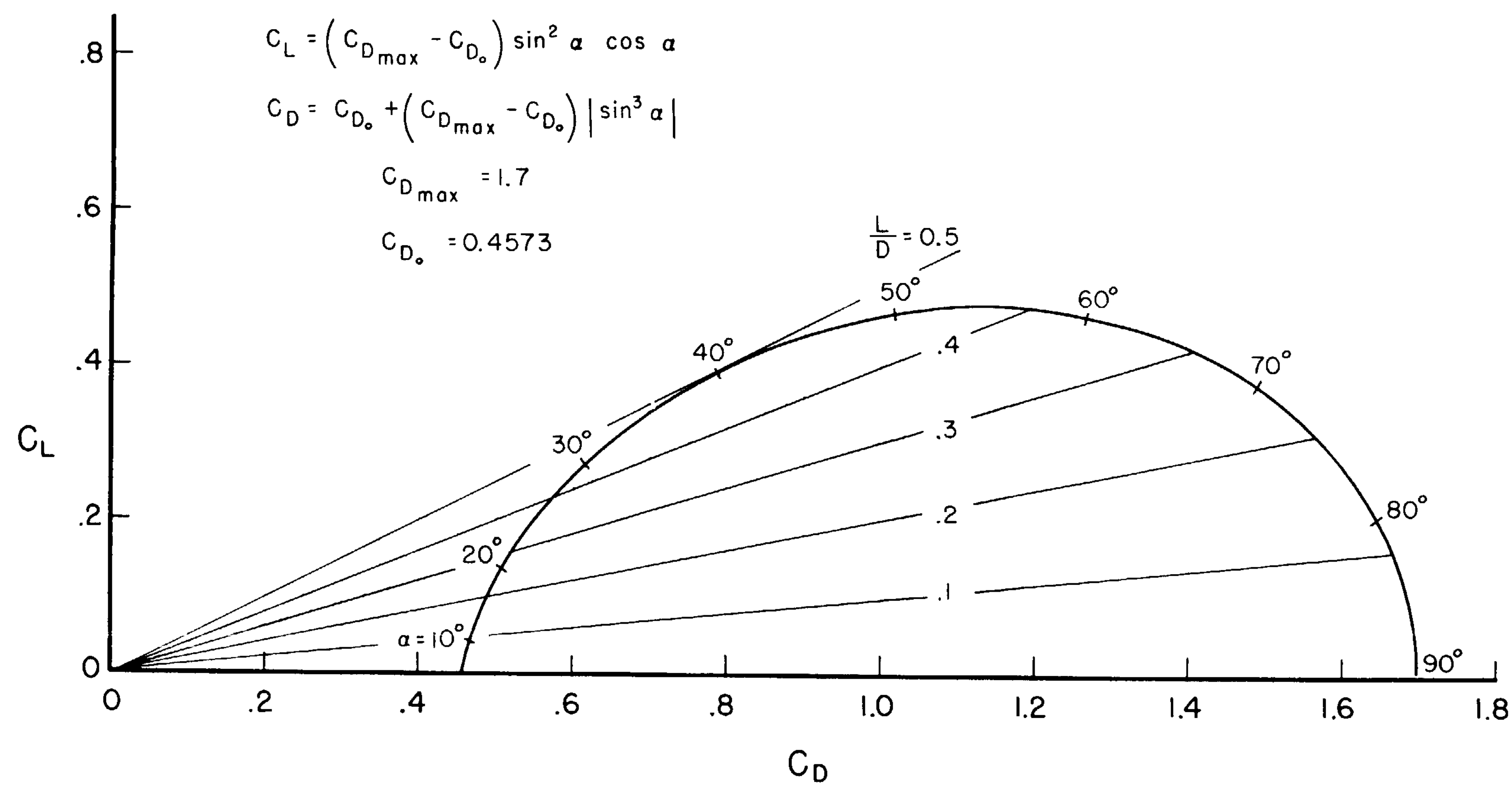

Figure 1.- Drag polar.

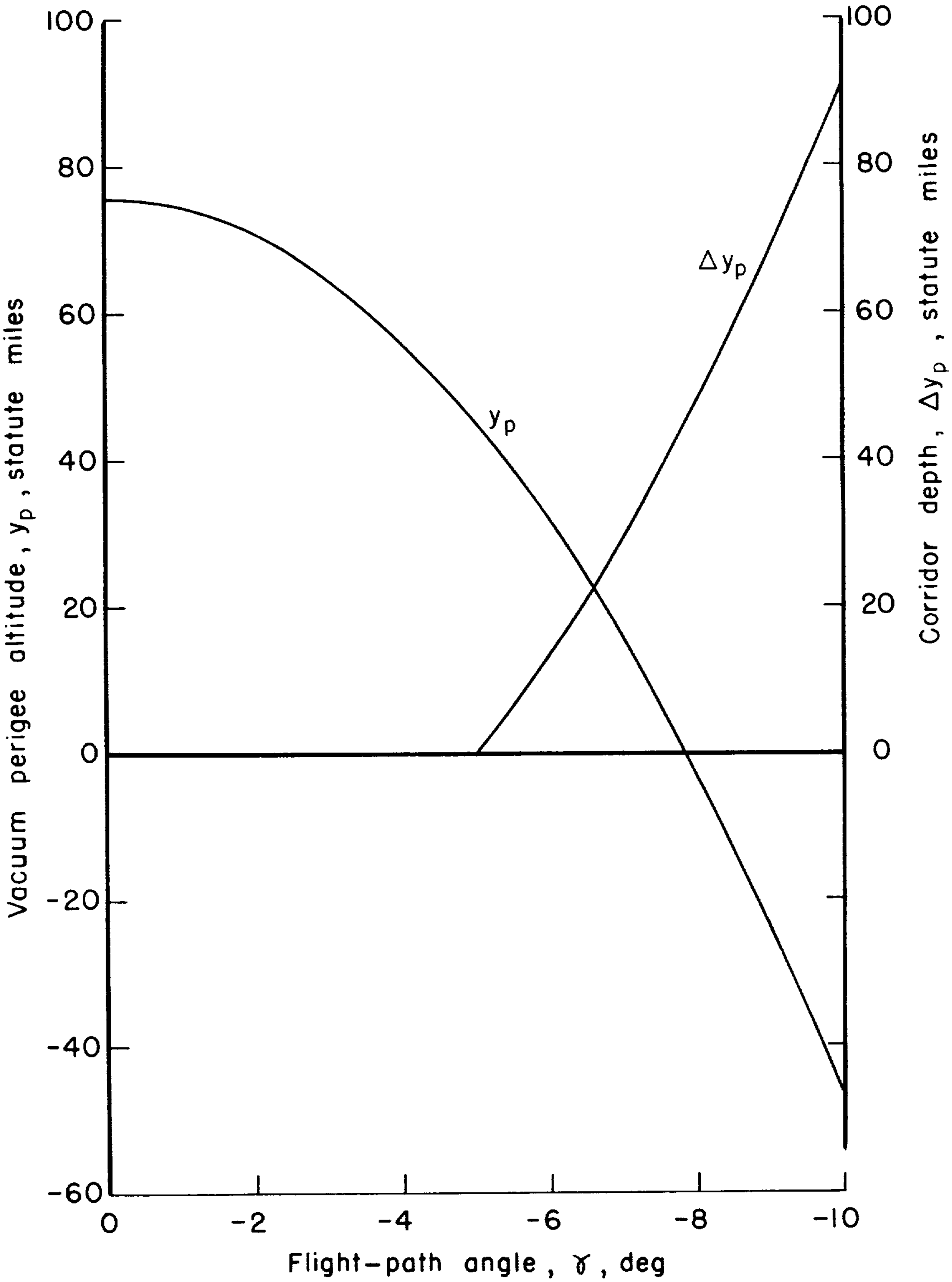

Figure 2.- Vacuum perigee altitude and corridor depth as functions of flight-path angle; y_i = 400,000 ft, v_i = 36,335 ft/sec.

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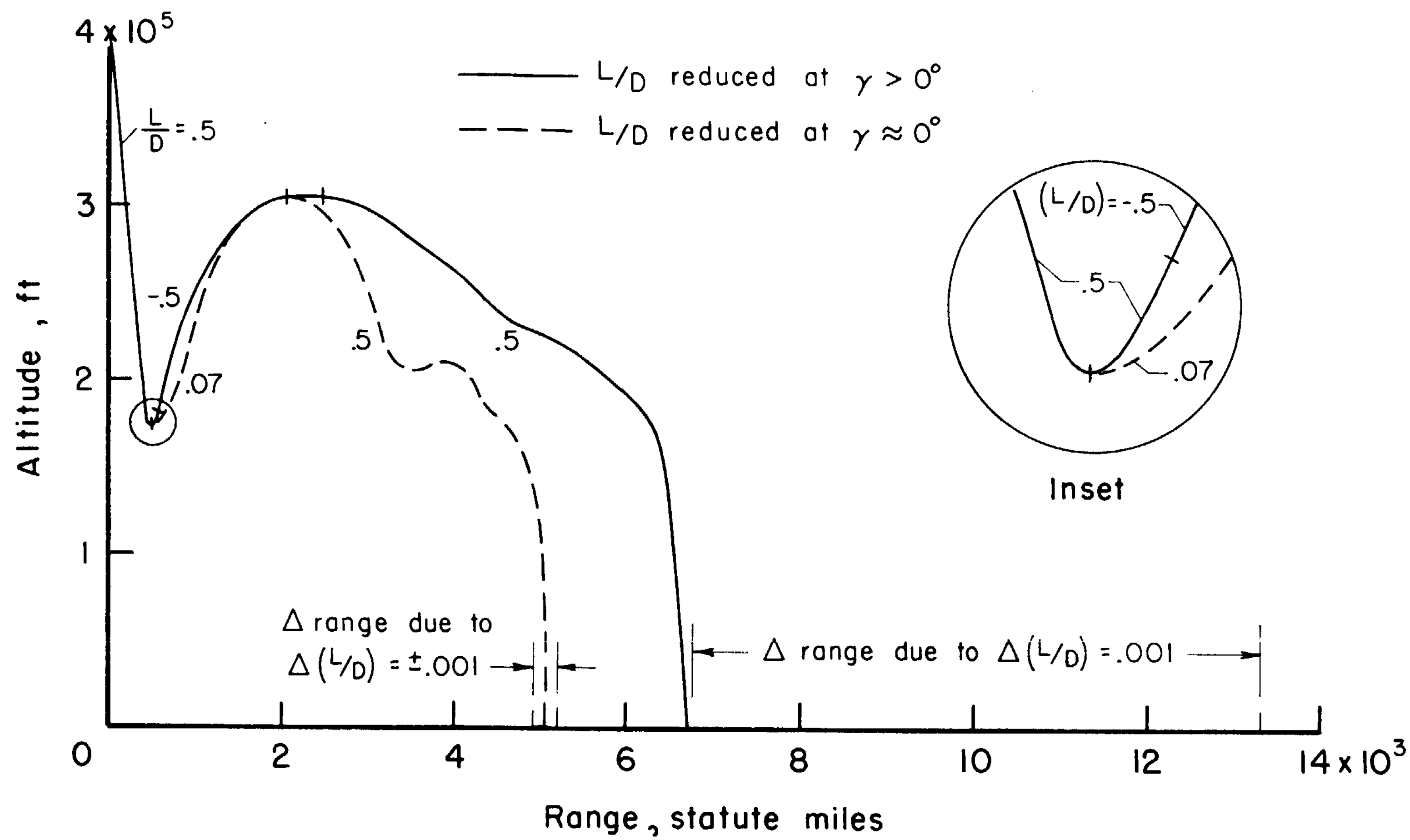


Figure 3.- Influence of type of maneuver and $\Delta(L/D)$ on range; $G_{max} = 10$.

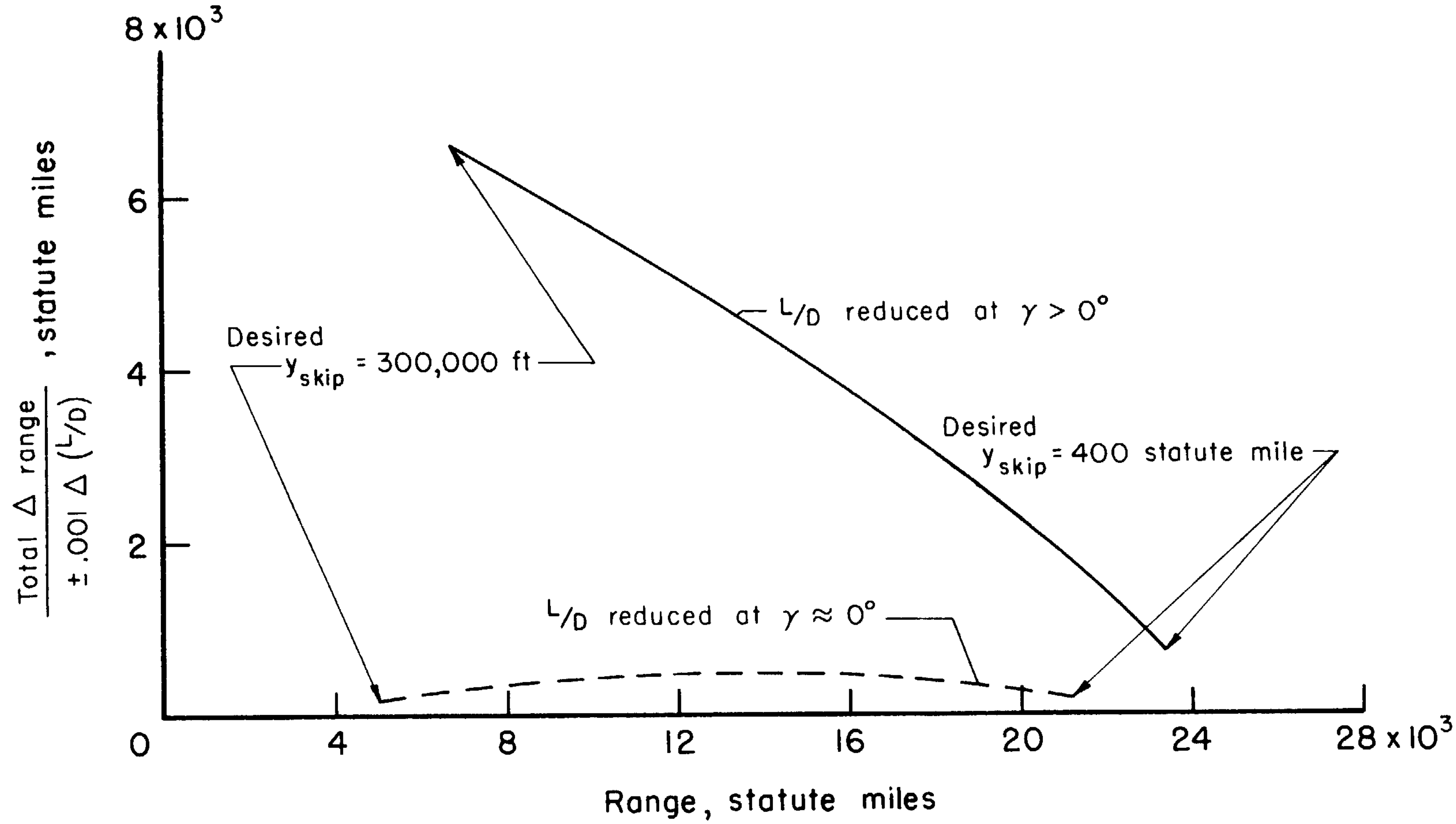
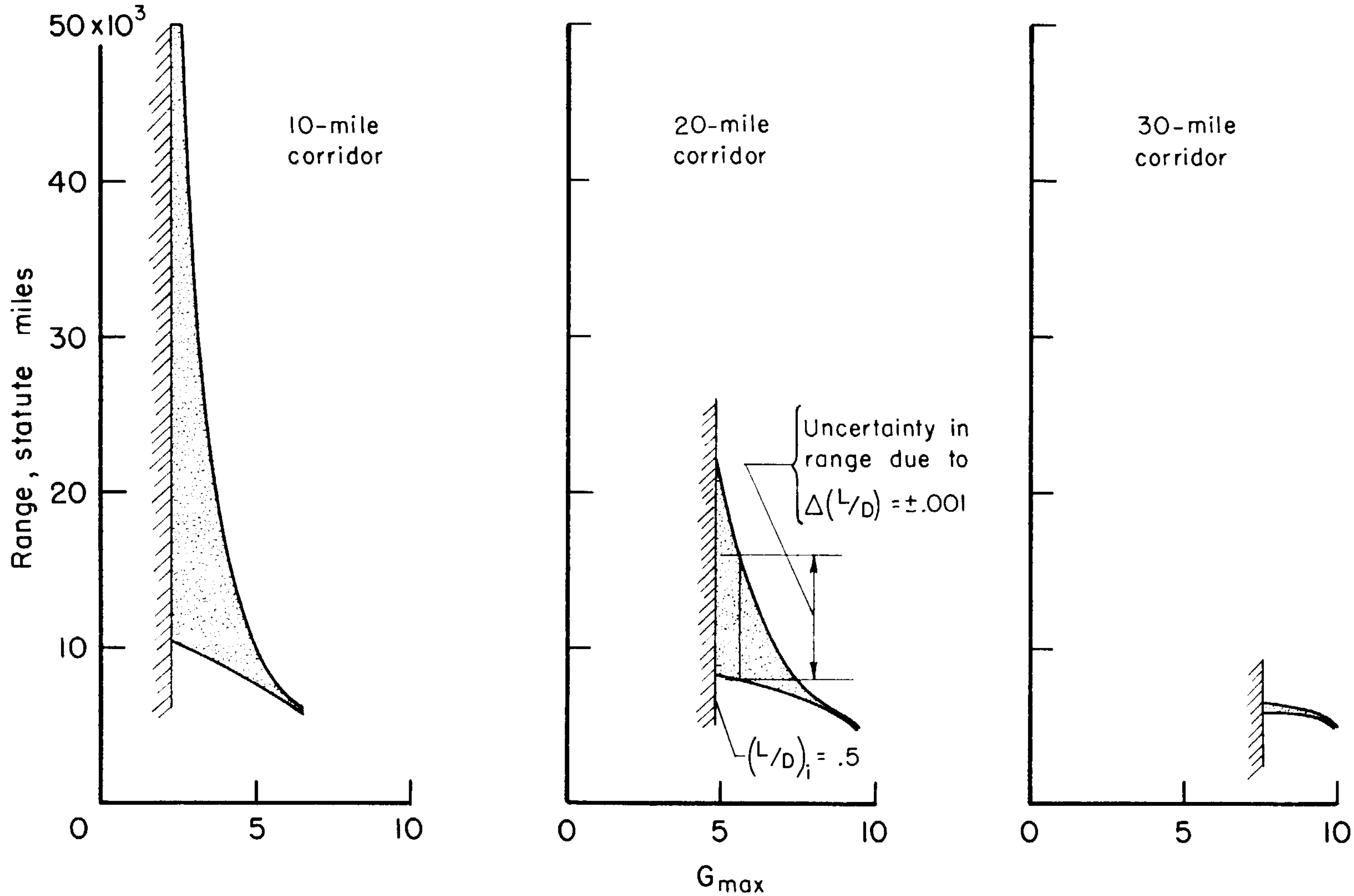
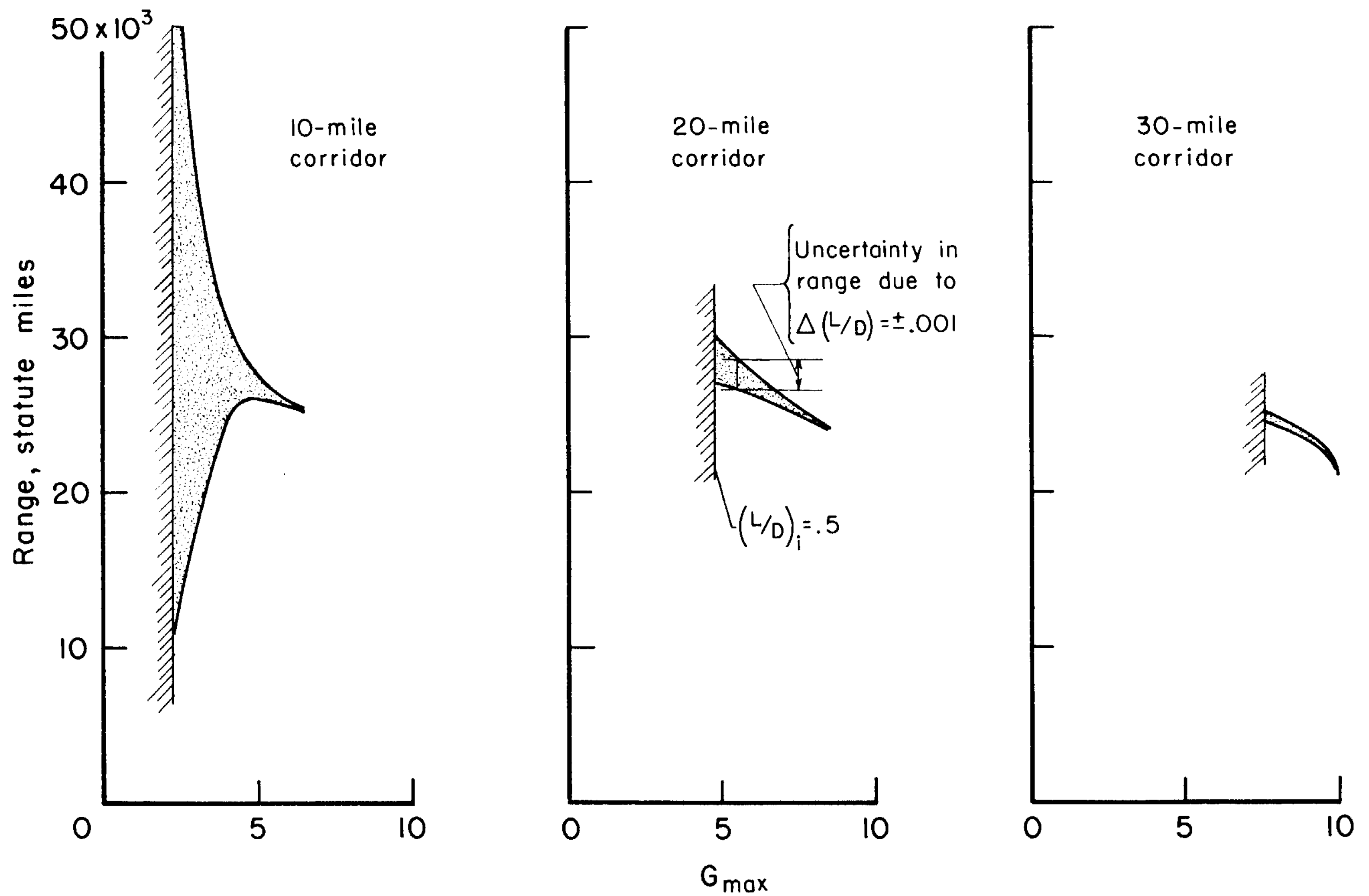


Figure 4.- Influence of type of maneuver and $\Delta(L/D)$ on uncertainty in range; $G_{max} = 10$, $(L/D)_i = 0.5$.



(a) Desired y_{skip} = 300,000 ft

Figure 5.- Influence of maximum deceleration on range.



(b) Desired y_{skip} = 400 statute miles

Figure 5.- Concluded.

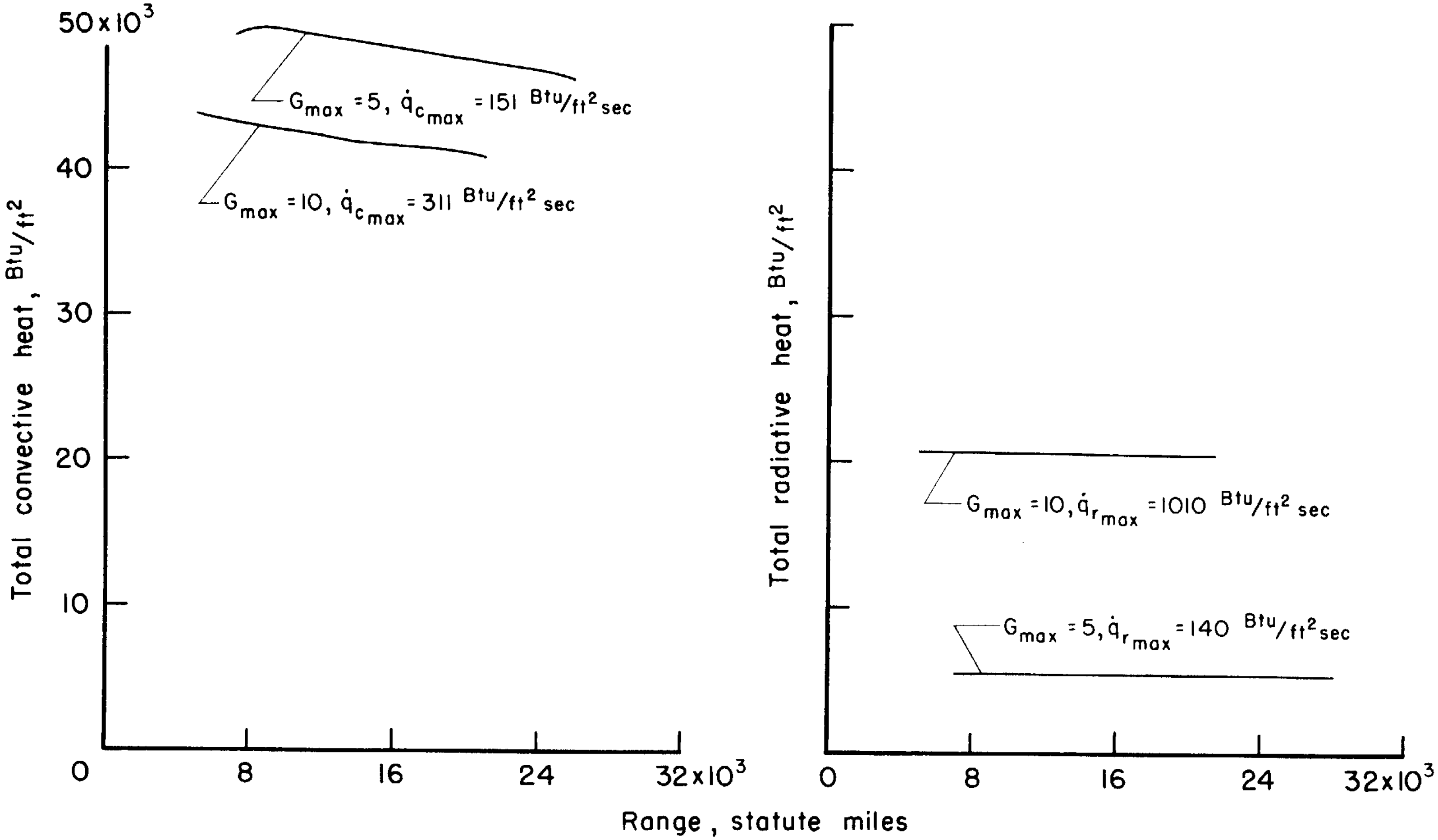


Figure 6.- Stagnation-point heating; R = 10 ft, m/A = 3 slugs/ft^2.

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